NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL MEMORANDUM

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DRAG REDUCTION BY SUCTION OF THE BOUNDARY LAYER SEPARATED BEHIND SHOCK WAVE FORMATION AT HIGH MACH NUMBERS

By B. Regenscheit

Translation

"Versuche zur Widerstandsverringerung eines Flugels bei hoher Machscher-Zahl durch Absaugung der hinter dem Gebiet unstetiger Verdichtung abgelosten Grenzschicht"

Deutsche Luftfahrtforschung, Forschungsbericht Nr. 1424

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SUMMARY: With an approach of the velocity of flight of a ship to the velocity of sound, there occurs a considerable increase of the drag. The reason for this must be found in the boundary-layer separation caused by formation of shock waves. It will be endeavored to reduce the drag increase by suction of the boundary layer.

Experimental results showed that drag increase may be considerably reduced by this method. It was, also, observed that, by suction, the position of shock waves can be altered to a considerable extent.

CONTENTS:

I. INTRODUCTION

Drag coefficient of a wing with an ordinary cross section has a tendency to increase considerably when

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*"Versuche zur Widerstandsverringerung eines Flügels bei hoher Machscher-Zahl durch Absaugung der hinter dem Gebiet unstetiger Verdichtung abgelösten Grenzschicht." Zentrale für wissenschaftliches Berichtswesen der Luftfahrtforschung des Generalluftzeugmeisters (ZWB) Berlin-Adlershof, Forschungsbericht Nr. 1424, July 1, 1941.

the velocity of the air flow approaches that of sound. The reason for this increase is understandable. During a very fast flight, there appears on the surface of the wing profile certain local velocities which exceed the velocity of sound. (This phenomenon may occur even then, when the lift coefficient c_a is equal to zero, and may be entirely due to the displacement of the air). In its further flow, the necessary air-flow retardation (the theoretical limit of velocity must be equal to $V = 0$ on the trailing edge of the wing) becomes discontinuous, which is in contrast with the usually continuous air-flow phenomena in incompressible air.

The discontinuity of retardation causes an appearance of a series of shock waves, creating a condition of a sudden decrease of velocity in a small interval of distance traveled, which in its turn causes a sudden change in density and pressure. These phenomena are, of course, not free from wasting of energy. This loss, however, does not constitute the principal cause of noticeable drag increase; rather it must be seen in a sudden increase of pressure (due to a formation of unstable shock waves) which forces a separation of the boundary layer from the wing's surface.

II. ARRANGEMENT FOR MEASUREMENT, MODELS, AND TEST PROCEDURE

The high-speed wind tunnel of the AVA (open-jet wind tunnel) was at disposal for the measurements. It has a test cross section 110 by 110 millimeters. O. Walchner (1) gave a description of such a wind tunnel with a slightly smaller jet cross section. A low pressure chamber of 40-meter^3 volume which was connected with the suction slot of the wing by a duct was used for increasing the suction quantity. The flow observations were carried out by means of the well-known method of schlieren optics (2).

The wake behind the wing section was measured with the aid of a Prandtl tube in order to determine the drag. The total test arrangement is shown in figure 1.

The model wings had for wing section a digonous circular-arc section of 17 percent thickness with rounded nose.

Figures 2 and 3 show the investigated wing sections. Suction slots in the direction of the flow were provided for a further wing (4) according to Professor Betz' suggestion.

The investigations were carried out for only one angle of attack ($\alpha = 0^\circ$), but for different Mach-numbers and suction quantities. The first part of the investigation was limited to the observation of the suction effect in the schlieren photograph. The suction slot was cut in only on the side which was to be observed for this part of the investigation. In the second part the wake behind the wing was measured point by point by means of a Prandtl tube. Now the suction slots were contrived on both sides.

III. EVALUATION OF THE RESULTS OBTAINED BY MEASUREMENTS

Velocity of the air flow u and its ratio to the velocity of sound a $\left(\text{Mach number} = \frac{u}{a}\right)$ were obtained in the usual manner (1), (3). The evaluation of results obtained by measurements of the wake was done with the use of formula:

$$W = \int_N \frac{2K}{K-1} p_p \sqrt{\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - 1} \left\{ \sqrt{\left(\frac{p_D g_p}{p_p g_D}\right)^{\frac{K-1}{K}} \left[\left(\frac{g_D}{p_D}\right)^{\frac{K-1}{K}} - 1\right]} - \sqrt{\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - \left(\frac{p_D}{p_p}\right)^{\frac{K-1}{K}}} \right\} dy_p$$

giving all necessary data for calculation of drag coefficient.

The derivation of this formula is given in the appendix.

The symbols encountered in the formula signify the following quantities:

g_D total pressure behind the nozzle outlet

p_D total pressure behind the nozzle outlet[1]

g_p total pressure measured in the Prandtl tube

p_p static pressure measured in the Prandtl tube

A correction of values obtained with the use of the Prandtl tube was not necessary, because, as it was proved by O. Walchner (1), the errors due to suction pressure indicator were very small for the condition of yaw angle $\varphi = 0$, even when the Mach number was as high as $M = 0.9$. Our measurements were rather rough.

For more accurate measurements, the suction apparatus was provided with a slide valve calibrated to register the change of pressure due to the suction of the air per second. With this adjustment of the slide valve, the pressure diminution in the suction box was determined for a period of about 30 seconds. It was found that, during this period, the pressure was diminished by 1 to 2 percent of the initial suction pressure.

The volume of sucked out air was found to be equal to $Q = \frac{40p}{760t}$ which corresponds to the following measured values:

40 volume of the suction box expressed in cubic meters, with a suction pressure $b = 760$ mm Hg

760 normal atmospheric pressure in mm Hg = 10.333 m H_2O

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[1]This is an obvious error in the German original; as p indicates static pressure, p_D = static pressure behind the nozzle outlet.

p increase of pressure per a time unit, in millimeter of mercury

t time, in seconds

The obtained Q value was used for determination of the velocity of the air flow and for that of c_Q-coefficient on the wing surface.

IV. RESULTS OF EXPERIMENTATION

Experiments connected with the study of the air flow showed that boundary layer can be influenced very strongly by suction.

Figures 5 and 6 illustrate the phenomena of the air flow.

Locations of suction slits are shown by arrows. Only one side of the wing is shown, the other is covered by suction apparatus.

Figure 5 shows the results observed with a suction slit cut out on the 70-percent point of the wing chord. When suction is not used ($c_Q = 0$), the first shock wave appears at the 50-percent point of the chord length (approximately); at 70-percent point, there begins another shock wave, whose direction opposes slightly that of the air flow. At 80-percent point, a third shock wave is seen.

When suction is used, a considerable change in the schlieren picture of the air flow is quite noticeable, even when the sucked-out air quantity coefficient c_{QE} is as small as $c_{QE} = 0.0024$. (The subscript E indicates that suction is used only on one side of the wing; when it is used on both sides c_Q-symbol is used, and is approximately equal to $c_Q = 0.0048$ when c_{QE} is equal to 0.0024.) In this case, the first shock wave does not begin on the wing's surface, but is formed in the free air flow above. The boundary layer, which has been definitely separated when c_Q was equal to zero ($c_Q = 0$), now adheres to the wing's surface to the very suction slit. The second shock wave is almost vertical, and the third is defined very feebly.

When coefficient c_{QE} reaches 0.0049 value, the first shock wave is very weak; the second wave is noticeably moved aft from the suction slit and is bent in the direction of the air flow; the third wave practically disappeared. The boundary layer is visible; in the neighborhood of the slit it becomes gradually thinner, as it is sucked out.

Figure 6 is a study of the air-flow conditions when the suction slit is cut out ahead of the shock waves. In this case, the slit was made at the 30-percent point of the wing's chord. When suction is not used, two distinctly outlined shock waves appear along the wing surface. The first, at (approximately) the 30-percent point of the chord length, is caused by the interference of the suction slit. The second wave appears automatically at (approximately) the 60-percent point of the chord. Ahead of the second shock wave a feeble third wave is formed. Between the first and the second waves there is a definitely thick boundary layer.

When suction is used, the first shock wave recedes to the trailing edge of the slit and is very sharply outlined. Over the slit, there occurs a change in the bending of the first shock wave. The second shock wave is also more pronouncedly bent in the direction of the air flow and, which is very noticeable, is displaced aft. The distances by which the beginning of the second shock wave was shifted on the surface of the profile when suction was applied is shown in figure 6.

The boundary layer between the first and the second shock waves became much thinner. A very feeble condensation line is seen between the top of the first shock wave and the bottom of the second.

A comparison between figures 5 and 6 gives an impression that the position of the shock waves is more definite in the last case. Furthermore, boundary-layer separation, as seen in figure 5, without the use of suction, does not appear in figure 6, which was taken when suction was applied.

Measurements of the wake were taken with the use of two diagonally cut slits and one slit cut in the direction

of the wing's length. The results of these measurements are shown in figures 7 and 8. A slanting slit at 70-percent of the wing chord was found to be superior to two other arrangements for Mach number $M = 0.9$. When c_Q was as low as 0.004, the drag diminution became apparent.

A wing with a slanting slit at 85-percent of the chord had a slightly greater drag than the wing with a slit cut out at 75-percent of the chord, when Mach number M was equal to 0.9 ($M = 0.9$), and $c_Q = 0$. The curves representing the drag for both arrangements (slit cut out at 85-percent and 70-percent points of the chord) are similar, when suction is used ($c_Q = 0$); the similarity begins with a rather larger quantity of sucked-out air.

Measurements with a slit cut out in the longitudinal direction are, generally speaking, convenient for larger drag values. In this case, a true drag value could not be measured. In order to obtain an approximately correct drag value, the drag should be measured in many points along the wing span, and the average of the results taken. Our experimental apparatus was not adapted for such a process of measurement.

It appears that a wing section with a very high drag had been chosen for this experimentation.

V. CONCLUSION

The problem of the present report consisted in proving that the drag of the wing profile appearing with high velocity of the air flow can be diminished by the use of suction producing devices.

Experiments performed with simple implements showed that with a larger quantity of sucked-out air, there occurred a considerable diminution of the drag. Among different arrangements of suction apparatus, the most favorable was that having the suction slit at the 70-percent point of the wing chord ($0.7l$). From schlieren photographs of the air flow, it may be assumed that suction slit placed ahead of the region

of shock wave formation is also not entirely ineffective with regard to drag diminution. Such arrangement was not used for testing the wake; an arrangement having one suction slit before and the other behind the region of shock wave formation was not used also. During the performance of the experiment efficiency of suction apparatus was very high with respect to the quantity of sucked-out air and the losses in the suction conduits. Therefore, a direct use of results obtained in this work should not be entirely possible in the airplane technique.

Further experiments must be performed for an investigation of this interesting physical phenomenon. This research will, perhaps, provide the possibility of obtaining such results which could be used in actual practice.

I should like to thank Mr. Ludwieg for his valuable assistance in performance of this work.

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Goodyear Aircraft Corporation

VI. APPENDIX

DERIVATION OF FORMULA FOR DETERMINATION OF DRAG FROM MEASUREMENTS OF WAKE AT HIGH VELOCITIES OF THE AIR FLOW

A formula for drag evaluation was derived on the basis of equations given by Kramer and Doetsch on the one hand, and Jones on the other hand, (4), (5).

According to these equations:

$$W = b \int_N \rho_1 U_1 \left(U_0 - U_2\right) dy_1 \qquad \text{kilograms}$$

where: U_1 and ρ_1 are, respectively, the velocity and the density in the cross section under consideration, U_0 is the wind velocity far ahead of the wing, and U_2 is the wind velocity far behind thw wing, b is the span length of the wing.

In the wind tunnel the following conditions were created:

$U_0 = U_D$ wind velocity directly behind the nozzle

$U_1 = U_P$ wind velocity measured by Prandtl tube directly behind the wing

For the evaluation of the velocity of the air flow, the following equations were obtained by transforming the Saint-Venant relationship:

$$U_D = \sqrt{\frac{2K}{K-1}\left(\frac{p_D}{\rho_D}\right)\left[\left(\frac{g_D}{p_D}\right)^{\frac{K-1}{K}} - 1\right]} \qquad \text{where: } g = \text{total pressure}$$

$$U_P = \sqrt{\frac{2K}{K-1}\left(\frac{p_p}{\rho_p}\right)\left[\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - 1\right]} \qquad \text{and } p = \text{static pressure}$$

U_2 can be calculated from U_P-equation assuming that there is no loss of energy between the two points of observation, and that the density is changing adiabatically with the static pressure. Assuming that the pressure at the point 2 is equal to P_D, we obtain:

$$\frac{U_P^2}{2} + \frac{K}{K-1}\frac{p_p}{\rho_P} = \frac{U_2^2}{2} + \frac{K}{K-1}\frac{P_D}{\rho_2}$$

$$\rho_2 = \rho_P\left(\frac{P_D}{p_p}\right)^{1/K}$$

$$U_2 = \sqrt{\frac{2K}{K-1}\left(\frac{p_p}{\rho_P}\right)\left[\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - 1\right] + \frac{2K}{K-1}\left[\frac{p_p}{\rho_P} - \frac{P_D}{\rho_P\left(\frac{P_D}{p_p}\right)^{1/K}}\right]}$$

$$= \sqrt{\frac{2K}{K-1}\frac{p_p}{\rho_p}\left[\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - {\frac{P_D}{p_p}}^{\frac{K-1}{K}}\right]}$$

Therefore, the drag is:

$$W = b\int_N \frac{2K}{K-1}p_p\sqrt{\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - 1}\left\{\sqrt{\frac{P_D\rho_P}{\rho_D p_p}\left[\left(\frac{g_D}{P_D}\right)^{\frac{K-1}{K}} - 1\right]}\right.$$

$$\left. - \sqrt{{\frac{g_p}{p_p}}^{\frac{K-1}{K}} - \left(\frac{P_D}{p_p}\right)^{\frac{K-1}{K}}}\right\}dy_p$$

We must now find an expression for the ratio $\frac{\rho_P}{\rho_D}$.

With a constant air flow impeded by the drag but devoid of the loss of heat, the total energy remains unchanged. Instead of a dissipation of energy, there occurs its transformation; a retardation of the air along the surface of the wing produces a transformation of energy of motion into heat.

Without making any considerable error it may be assumed that there does not occur any loss of heat. Therefore, the energy equation between sections D and P can be written thus:

$$\frac{U_D^2}{2} + \frac{K}{K-1}\frac{p_D}{\rho_D} = \frac{U_P^2}{2} + \frac{K}{K-1}\frac{p_P}{\rho_P}$$

Therefore:

$$\frac{\rho_P}{\rho_D} = \frac{\frac{U_P^3 \rho_P}{2} + \frac{K}{K-1} p_P}{\frac{U_D^2}{2}\rho_D + \frac{K}{K-1} p_D}$$

$$= \frac{p_P\left[\left(\frac{g_P}{p_P}\right)^{\frac{K-1}{K}} - 1\right] + p_P}{p_D\left[\left(\frac{g_D}{p_D}\right)^{\frac{K-1}{K}} - 1\right] + p_D}$$

$$= \left(\frac{g_P}{g_D}\frac{p_D}{p_P}\right)^{\frac{K-1}{K}} \frac{p_P}{p_D}$$

When this value is introduced into the formula for drag we obtain:

$$W = b\int_N \frac{2K}{K-1} p_p \sqrt{\left[\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} -1\right]} \left\{\sqrt{\left(\frac{g_p}{g_D}\frac{p_D}{p_p}\right)^{\frac{K-1}{K}} \left[\left(\frac{g_D}{p_D}\right)^{\frac{K-1}{K}} -1\right]} - \sqrt{\left(\frac{g_p}{p_p}\right)^{\frac{K-1}{K}} - \left(\frac{p_D}{p_p}\right)^{\frac{K-1}{K}}}\right\} dy_p$$

Drag coefficient can be then determined from the dynamic pressure for any given velocity U_D and area of the wing $F = bl$ by formula

$$c_W = \frac{W}{lb\left\{\frac{K}{K-1} p_D \left[\left(\frac{g_D}{p_D}\right)^{\frac{K-1}{K}} - 1\right]\right\}}$$

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1. Walchner, O.: The Effect of Compressibility on the Pressure Reading of a Prandtl Pitot Tube at Subsonic Flow Velocity. NACA TM No. 917, 1939.

2. Prandtl, L.: Abriss der Strömungslehre, p. 195, Verlag Friedr. Vieweg und Sohn, Braunschweig 1935.

3. Betz, A.: Hütte Band I, Gasdynamik, p. 413.

4. Doetsch, H.: Ergänzende Mitteilungen zum Bericht Profilwiderstandsmessungen im grossen Windkanal der DVL, Lufo Bd. 14, 1937, Heft 7, p. 367.

5. Jones, B. M.: Measurements of Profile Drag by the Pitot Traverse Method. ARC Rep. Nr. 1688, London 1936.

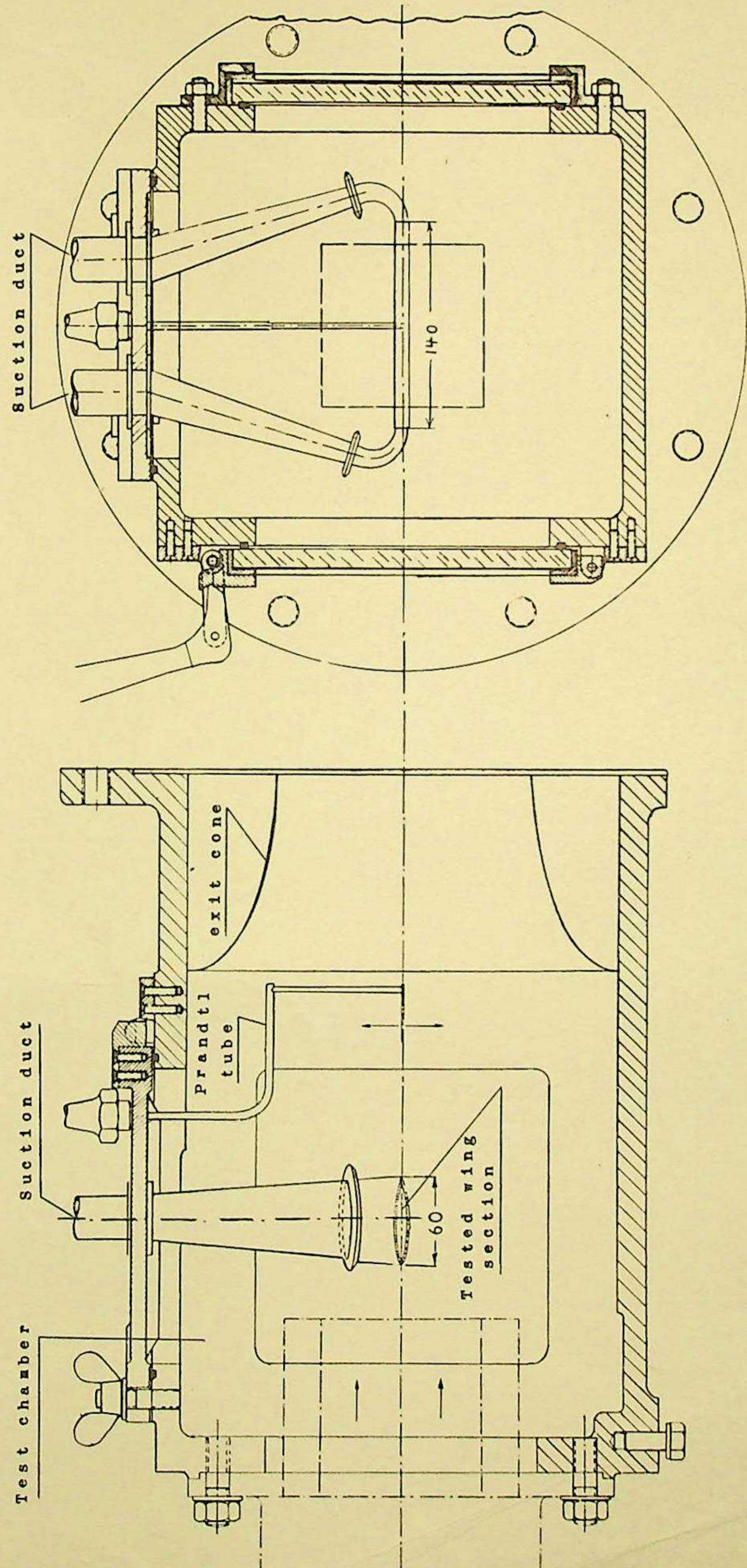

Figure 1.- Prandtl's tube used for measurement of wake behind the wing. Suction-influence on the drag at high velocities of flight.

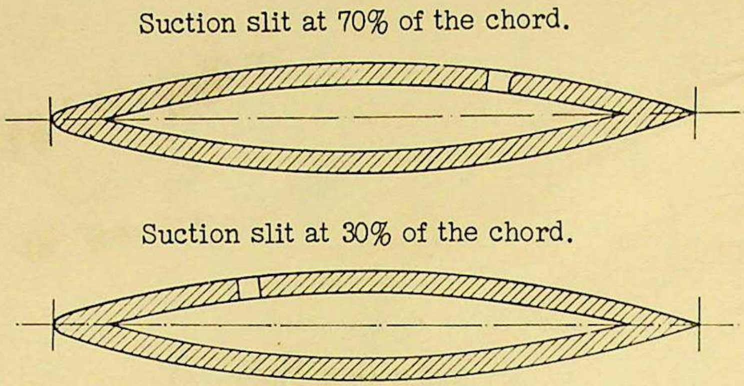

Figure 2.- Cross-section of the wing used for air-flow observations.

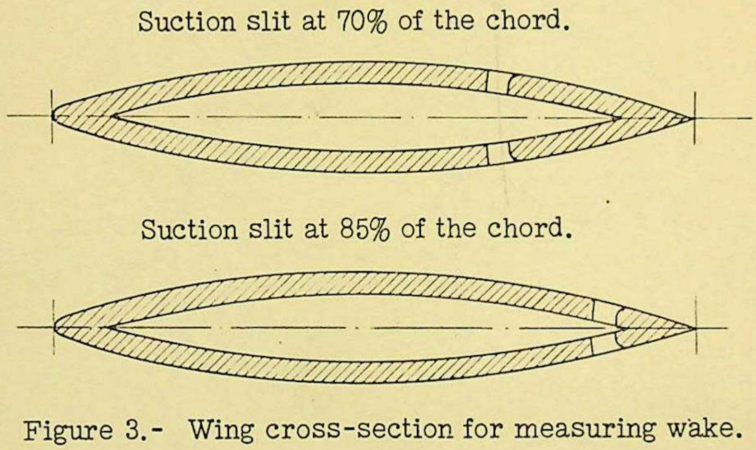

Figure 3.- Wing cross-section for measuring wake.

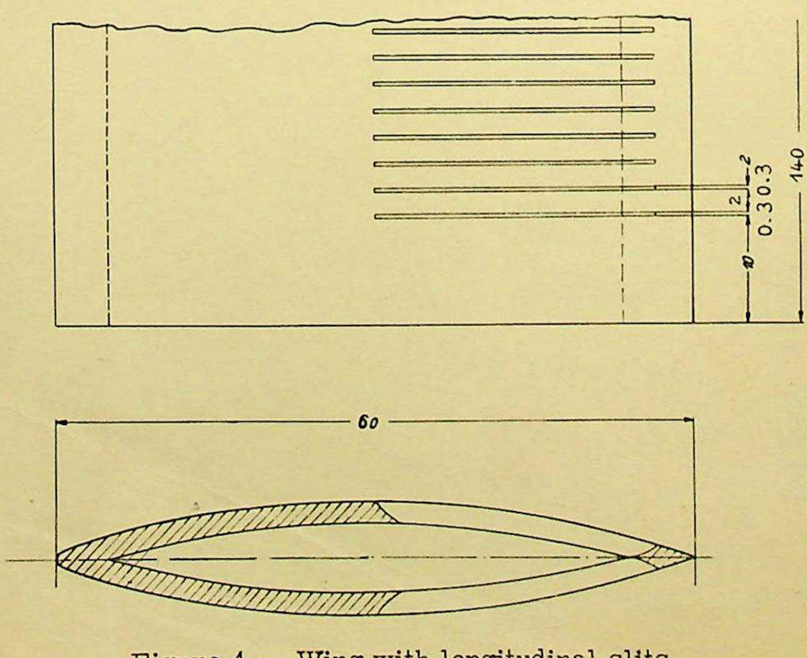

Figure 4.- Wing with longitudinal slits.

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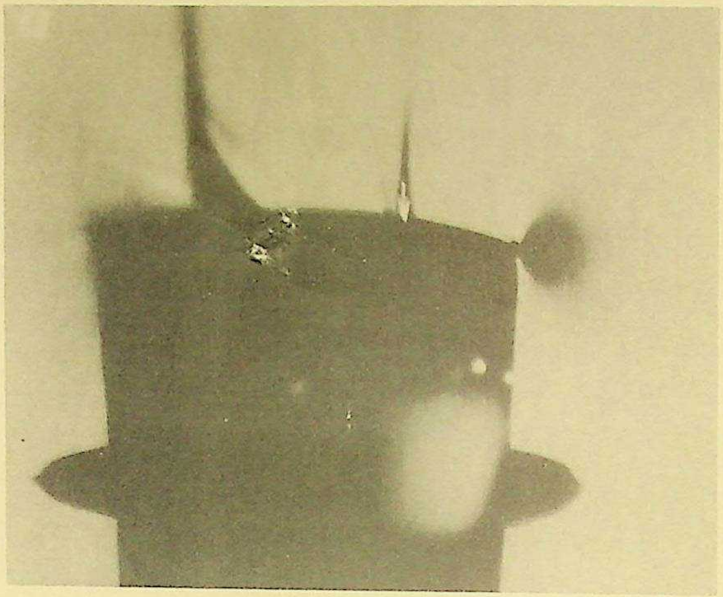
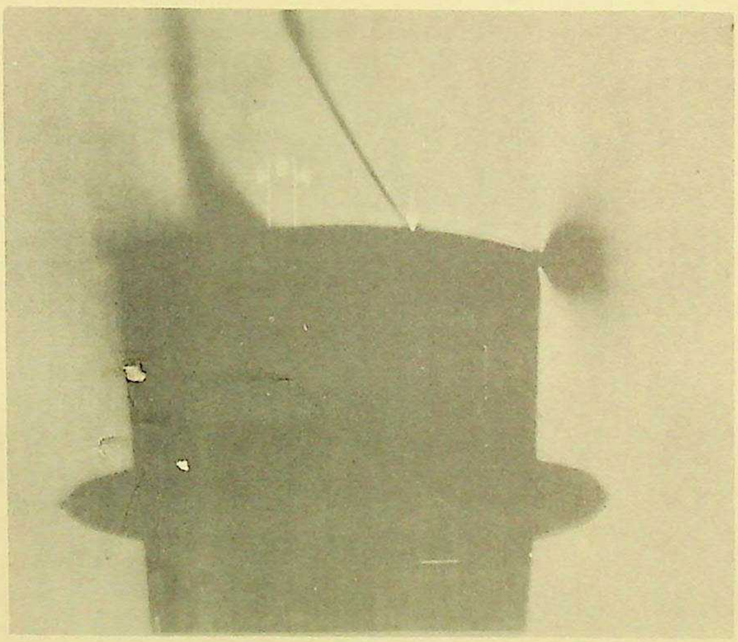


$C_{Q_E} = 0$



$C_{Q_E} = .0049$

air flow direction ←

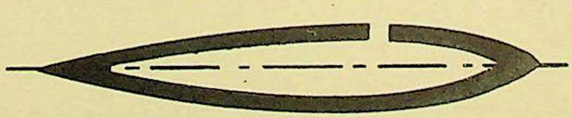
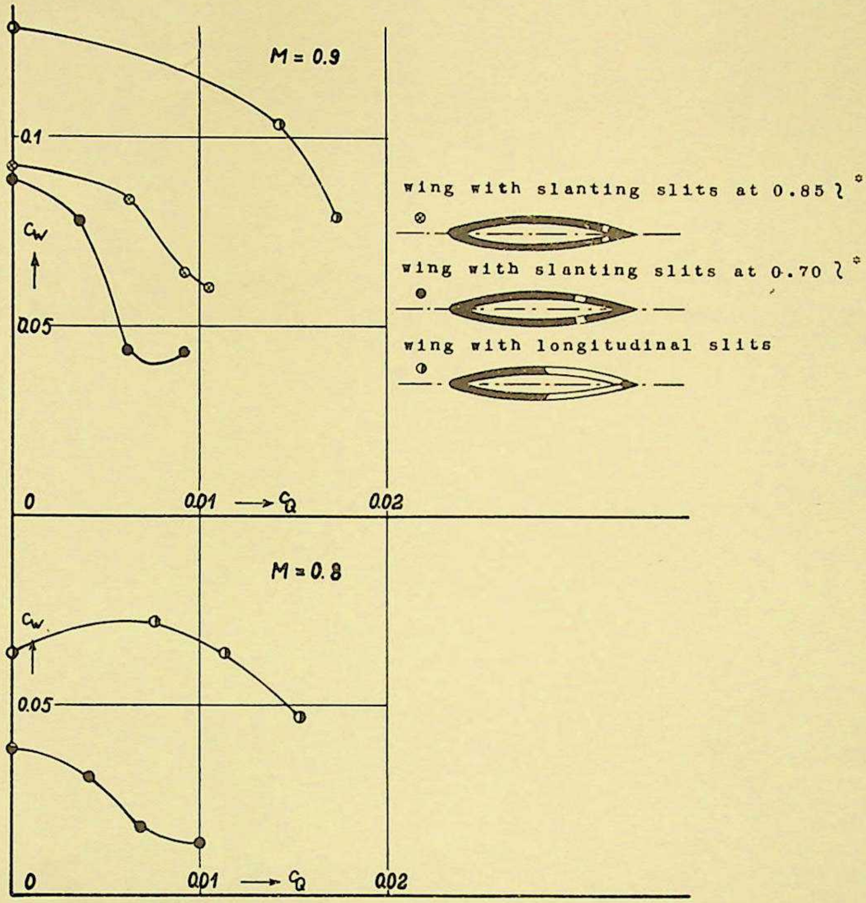


Figure 6.- Wing with a suction slit at 30% of the chord, M = 95. The suction refers only to one side of the wing. Arrow on Schlieren picture indicates the position of the slit. Distance S indicates the shift of the second shock wave produced by suction.



Figures 7 and 8.

Variation of drag coefficient with different position of suction slit.

*Translator's note: *l is the length of the wing-chord.